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Delbert E. Robison

Interim Report No. 3
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# PURDUE UNIVERSITY AND PURDUE RESEARCH FOUNDATION Lafayette, Indiana

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BOOKST LABORATORY

May, 1953

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EXPERIMENTAL ROCKET NOTOR PERFORMANCE VITH WHITE FUNING NITRIC ACID AND JP-4 AT 1000 PSIA COMBUSTION PRESSURE

BY

#### Delbert E. Robison

#### .. I. INTRODUCTION

The research program, of which the experiments reported herein form a part, is concerned with obtaining experimental data pertinent to the performance and general characteristics of a rocket motor operating at high combustion pressures using whits fuming nitric acid (VIIIA) and jet propulsion fuel (JP-4) as the propellants. Some of the experiments were conducted using JP-3 as the fuel, and some using JP-4.

In addition to investigating rocket motor performance, average heat transfer rates were measured for the combustion chamber and the nossle.

At least 95 per cent of the theoretical (fregen composition) values of specific impulse for the propellants was obtained at all mixture ratios by utilizing a triplet type injector in conjunction with a turbulence ring similar to the one described in reference 1.

# II. PERFORMANCE AND HNAT TRANSFER AT 1000 PSIA COMBUSTION PRESSURE

Figure 1 presents the specific impulse I<sub>sp</sub>, the thrust coefficient O<sub>f</sub>, and the characteristic velocity O<sup>6</sup>, uncorrected for the heat transferred to the cooling water, as functions of the mixture ratio.

The theoretical values for WFMA and n-octane, based on frozen equilibrium composition (references 4 and 5) are also presented for comparison. It is seen that the maximum values of specific impulse, thrust coefficient,

<sup>1.</sup> Mixture ratio = 0/F = oxidizer weight flow rate

fuel weight flow rate

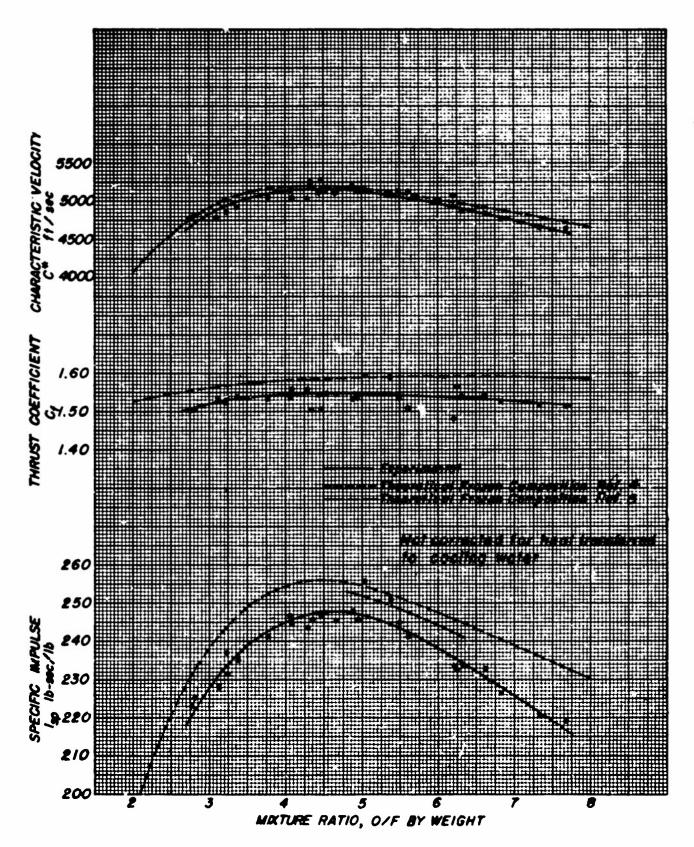


Fig. 1. Performance versus Mixture Ratio for WFNA and JP-4 at 1000 psia Combustion Pressure

and characteristic velocity occur at a mixture ratio of appreximately 4.6. Their respective values are  $I_{sp} = 248$  lb sec per lb,  $C_f = 1.55$ , and  $C_f = 5150$  ft per sec.

Figure 2 presents the experimental specific impulse corrected for the heat transferred to the cooling water,  $I_{\rm spc}$ , and is compared with the theoretical values mentioned above. A smooth curve drawn through the experimental points indicates that the maximum corrected specific impulse is 256 lb-sec per lb and occurs at a mixture ratio of 4.6. In the mixture ratio range 4 to 5.5 there is little difference between the experimental values of  $I_{\rm sp}$  and the theoretical values.

In the mixture ratio ranges 2.8 to 4 and 5.5 to 7.8 the experimental values of I are smaller than the theoretical values, and the difference between the two values tends to increase as the mixture ratio departs from the optimum value of 4.6. Since one injector, in conjunction with a turbulence ring, was employed over the entire mixture ratio range of 2.8 to 7.8, the differential pressures acting on the exidizer and fuel orifices differed widely at the extreme limits of the mixture ratio range. For example, the pressure drop across the fuel orifices decreased from 200 psi for a mixture ratio of 3 to 50 psi for a mixture ratio of 7, while the corresponding pressure drops for the exidizer orifices increased from 120 psi to 245 psi. It seems reasonable to assume from previous work conducted at this laboratory that if the experiments had been conducted with several injectors, each designed for the particular mixture ratio being studied, the difference between the corrected experimental values of I and the theoretical values would be much smaller.

When the rocket motor was operated with rich mixture ratios below 3, the combustion tended to be rough, but no serious instability was encountered. "Smooth" sounding, stable operation was obtained in the mixture ratio range of 3 to 5. With mixture ratios leaner than 5.5, unstable combustion was generally encountered, but not always. Stable operation was obtained at leaner mixture ratios with the first 6-point

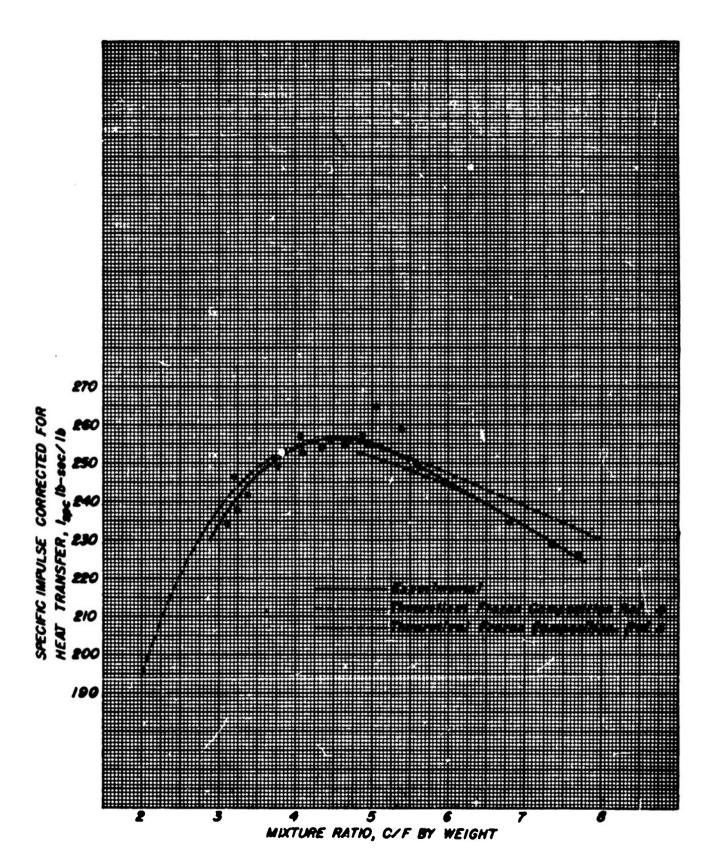


Fig. 2 Specific Impulse Corrected for Heat Transfer vs Mixture Ratio for WFNA and JP-4 at 1000 psia Combustion Pressure.

injector (2:1-65-1) employed than with the second (2:1-65-2). The enly difference between the two injectors was that the pressure drops acreed the second was slightly lower.

Figure 3 presents the average heat transfer fluxes for the combustion chamber, the nossie, and the over-all heat transfer flux for the rocket meter. The ever-all heat transfer flux is defined as the ratio of the sum of the nossie and the combustion chamber heat transfer rates to the combined inside surface area of the two components. The ever-all heat transfer flux does not include the heat transferred (approximately 6 per cent of the total) to the turbulence ring.

Although the data presented in Fig. 3 are semewhat scattered, the dependency of heat transfer on mixture ratio is quite evident. The trend of the data indicates that the maximum value for the average heat transfer flux for the nextle was 5.3 B/sec-in<sup>2</sup> and occurred at a mixture ratio of 4.5. The average heat transfer flux for the combustion chamber was a maximum at a mixture ratio of 4.5, having the value 4.2 B/sec-in<sup>2</sup>.

The detailed experimental data obtained at 1000-psis combustion pressure are tabulated :- Table 1, Appendix A.

#### III. EFFECT OF COMBUSTION PRESSURE ON PERFORMANCE AND HEAT TRANSFER

It is of interest to compare the performance of rocket motors operating at 1000-psia combustion pressure with that of motors operating at combustion pressures of 300, 500, and 700 psia. The performance of WFMA and JP-3 for the latter combustion pressures is presented in references 1 and 2, and was obtained under the same program as the 1000-psia data reported herein. Geometrically similar motors, each developing 500 pounds thrust, were employed throughout the program.

The effect of combustion pressure on the corrected specific impulse for mixture ratios of 3.0, 4.5, and 6.0 is shown in Fig. 4. Except for p<sub>G</sub> = 1000 psia and 0/F=4.5, the experimental values are all lower than the theoretical values, the average differences being 1.4 per cent for a mixture ratio of 4.5, and 3.6 per cent for mixture ratios of 3.0 and 6.0. The experimental specific impulse may not represent the maximum values that can be attained for mixture ratios of 3 and 6. They do, however, represent the performance that can be expected if one injector is employed for

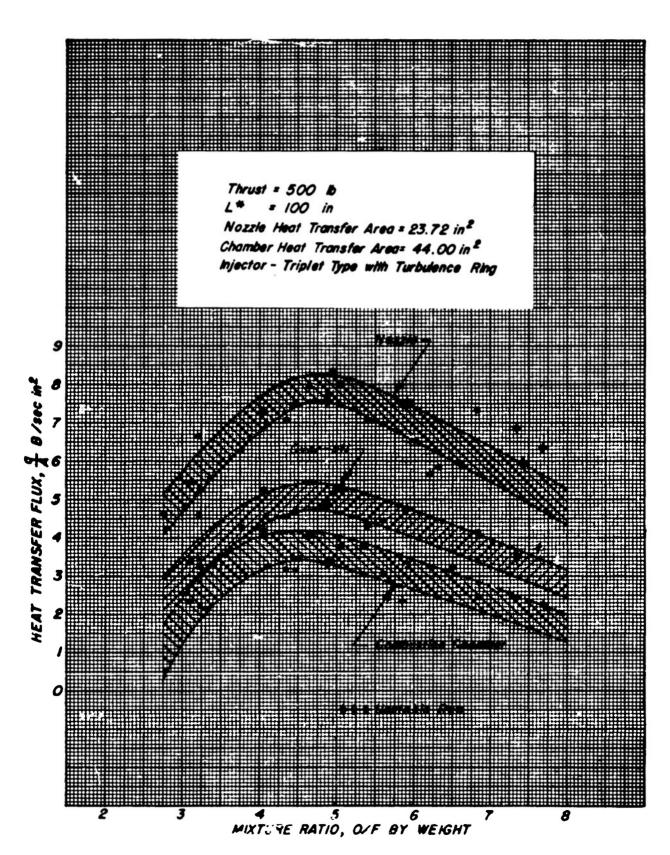


Fig. 3 Heat Transfer Flux vs Mixture Ratio For WFNA and JP-4 at 1000 psia Combustion Pressure

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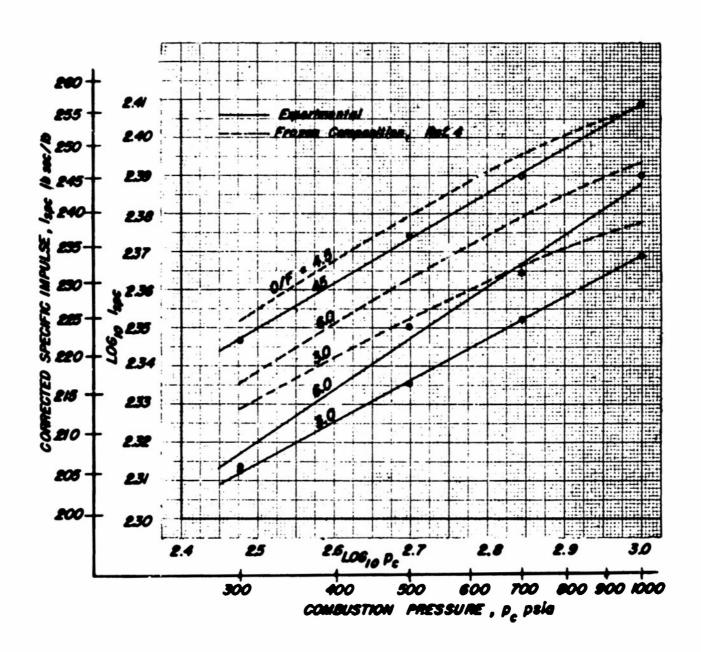


Fig. 4. Variation of Specific Impulse Corrected for Heat

Transfer with Combustion Pressure at Different

Mixture Ratios.

the mixture ratio range of 3 to 6.

The variation of the experimentally determined specific impulse with combustion pressure is represented closely by the exponential curve

$$(I_{\text{spc}})_1 = (I_{\text{spc}})_2 (\frac{p_{\text{cl}}}{p_{\text{cl}}})^m = 0p_{\text{c}}^m$$
 (1)

in which C is a constant and m is the slope of the experimental curve.

Both the exponent m and the constant C have different values for different mixture ratios.

Employing two points on each experimental curve shown in Fig. 4, the following equations were determined for the combustion pressure range of 300 to 1000 psia. For the optimum mixture ratio 4.5, the maximum csrrected specific impulse is given by

$$I_{spc} = 113.2 p_c^{0.1184} + 0.5 lb-sec/lb$$
 (2)

For the mixture ratio 3.0,

$$I_{epg} = 110.2 p_g^{0.1090} \stackrel{+}{-} 0.5 lb-sec/lb$$
 (3)

and for the mixture ratio 6.0

$$I_{end} = 95.91 p_0^{0.1353} + 1.7 lb-sec/lb$$
 (4)

Figure 5 shows the effect of combustion pressure on the heat transfer. The shaded areas represent the maxima and minima, due to scattering of the data for the peak heat transfer flux, for combustion pressures of 300, 500, 700, and 1000 psia. At each pressure, the maximum average heat transfer flux was obtained in the mixture ratio regime 4.5 to 5. The maximum heat transfer fluxes for the noszle and the combustion chamber at 1000 psia are both appreximately 3 times the respective values measured at 300 psia.

The experimentally determined heat transfer fluxes are represented within - 0.5 B.sec-in by the following linear equations:

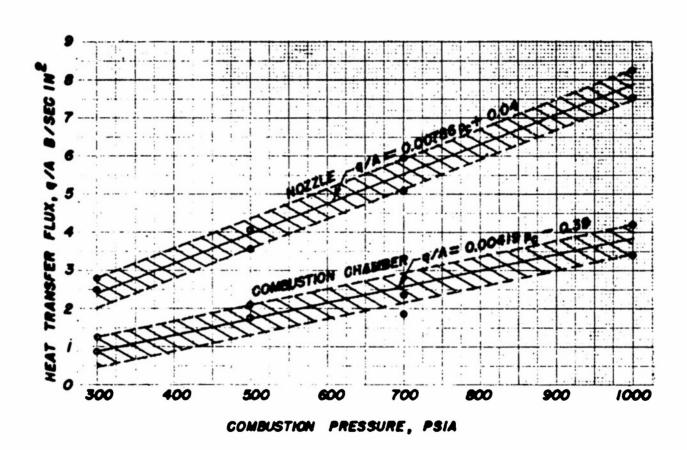


Fig. 5. Variation of Maximum Heat Transfer Flux with Combustion Pressure.

Hossle, 
$$q_n/A_n = 0.00786p_c + 0.04$$
 (5)

Chamber, 
$$q_a/A_c = 0.00419 p_0 - 0.39$$
 (6)

It is interesting to note that the nossle heat transfer flux is essentially directly proportional to the combustion pressure, and that within the scatter of the experimental data, the combustion chamber heat transfer flux could also be considered as being proportional to the combustion pressure.

During the course of the experimental work a short investigation was made of the suitability of available ceramic materials as liners for the combustion chamber and noszle of a 500 lb thrust rocket motor operating at 1000-psia. All of the ceramic liners failed in less than one second of operation, but the ceramic lined combustion chambers were undamaged after firing durations of 10 seconds. No effort was made, however, to establish the life of those liners.

#### IV. CONCLUSIONS

- 1. The maximum performance of a rocket motor operating on WFMA and JR-4 at 1000-psia combustion pressure and producing 500 pounds thrust occurs in the mixture ratio regime 4 to 5. Hore specifically, the trends of the experimental curves indicate that a maximum specific impulse of 256 lb-sec/lb can be obtained at a mixture ratio of 4.6. This is a 4.5 per cent increase above the maximum value obtained at 700-psia combustion pressure, and 15.4 per cent greater than the maximum value obtained at 300-psia combustion pressure.
- 2. The theoretical specific impulse based upon the frozen equilibrium compositions given in reference 4 closely approximates the experimental corrected values in the mixture ratio regime 4 to 5.5. For mixture ratios less than 4 and greater than 5.5, the experimental values were lower; e.g., 3 per cent less at a mixture ratio of 7.5, and 1.7 per cent less at a mixture ratio of 3.0. The departure of the experimental performance from the theoretical values for mixture ratios above and below the optimum range may have been due to poor mixing provided by the injector.

It may, therefore, be possible to improve the performance at the extreme mixture ration by employing an injector designed for the required mixture ratio.

3. For the combustion pressure range 300 to 1000-psia, with mixture ratios between 3 and 6, the specific impulse corrected for the heat transferred to the cooling water can be represented within 1 per cent by the exponential equation

in which 0 and m are constants which depend on the mixture ratio.

- 4. The maximum average heat transfer fluxes obtained at 1000psia combustion pressure for the negate and the combustion chamber are
  both approximately 3 times those obtained at 300 psia. Within the experimental error, the negate and combustion chamber heat transfer fluxes can
  be expressed as linear functions of combustion pressure, and are both almost directly proportional to the combustion pressure.
- 5. We difference between the performance of JR-4 and WFMA and that of JR-3 and WFMA was detected.
- 6. It appears that as good performance can be obtained with a 6-impingement point triplet injector as with a 12-impingement point triplet injector when either is employed in conjunction with a turbulence ring.
- 7. It is difficult to obtain stable combustion throughout the complete mixture ratio range of 2.8 to 7.8 with only one injector. Unstable combustion occurred more frequently for mixture ratios learns than 5.5. To operate over such a wide range of mixture ratios with one injector and still maintain supercritical pressure drops across the injector orifices may require differential pressure of the order of 300 to 400 psi at the extreme ands of the range. It is realized that the phenomenon of instability is not as yet well understood, and that the injector pressure drop and the quality of the mixing of the propellants after injection may

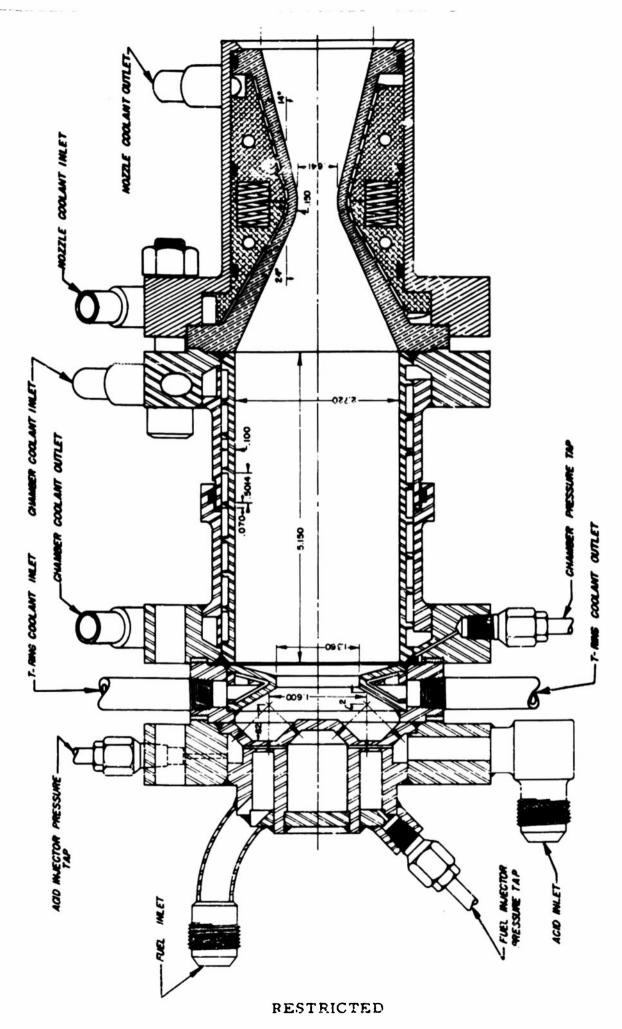
or may not be the principal cause of the combustion instability.

- at 1000-psia combustion pressure with a coolant flow rate of approximately 1 lb per sec, and a pressure drop of approximately 760 psi. It appears, however, that the copper nozzle will require some type of external surport, for example from the filler block, to prevent buckling.
- 9. None of the ceramic materials tested is satisfactory for the nossles of uncooled rocket motors which are to operate at maximum performance at 1000-psia combustion pressure.
- 10. It may be possible that ceramic liners may withstand the conditions prevalent in the combustion chamber, but more experimental information is required to establish that conclusion.

#### V. DESCRIPTION OF EXPERIMENTAL ROCKET MOTORS

Four rocket motors, each having an L\* of 100 and designed to produce 500 pounds thrust, were employed for the investigation conducted at 1000-psia combustion pressure. Three of the motors were uncooled and were utilised for obtaining starting-up characteristics and preliminary performance information. One of the uncooled motors consisted of a nia-frax combustion chamber and noszle comented in a mild steel retainer. The second uncooled motor comprised a "Borolite I" noszle and combustion chamber cemented in a mild steel retainer. The third uncooled motor consisted of a stainless steel combustion chamber and a solid copper nossle.

The fourth motor was water cooled and was the one employed for obtaining all of the performance and heat transfer data reported herein. Figure 6 is a cross-section through the experimental motor. Figure 7 is a photograph giving an exploded view of the motor assembly. The combustion chamber comprised an AISI type 347 stainless steel cylinder with helical grooves machined in the outer surface to form a helical coolant passage of rectangular cross section. Two combustion chambers of the latter design were employed during the investigation.



Cross Section of 1000-psia Water-Cooled Rocket Motor • FIG.

Three water-cooled copper nozzles were employed in the course of the investigation. One nozzle employed a helical coolant passage formed by silver soldering copper wire to the outer wall of the copper liner. That design proved to be unsatisfactory because the high coolant pressure deformed the wire thereby obstructing the coolant passage. In the two other copper nozzles the helical coolant passages were machined in their outer walls. Figure 8 illustrates the method employed for introducing the cooling water around the nozzle. The arrangement shown provides a uniform decrease in cross-sectional area of the coolant passage from the coolant inlet section to the wall of the copper nozzle, and in addition eliminates stagnation of the coolant at the inlet and outlet sections.

of the experiments; one had 12 impingement points, and two had 6 impingement points. The general configuration is illustrated in Fig. 6. Acid is ejected from the outer and inner circles of holes, while fuel is ejected from the circle of holes lecated between the two. All of the reported data were obtained with a triplet-type injector operating in conjunction with a turbulence ring. Each injector was designed to provide a pressure drop greater than 50 psi for the mixture ratio regime 3 to 7. The pressure drop versus flow rate characteristics for the turbulence ring and for each injector are presented in Figs. 9, 10, 11 and 12. The type of impingement and the resulting apray pattern produced with water by the 12-point injector is shown in Fig. 13. The photograph was taken before the injector was employed for a firing test. Figure 14 is a photograph of the spray pattern produced by the same injector when equipped with a turbulence ring.

#### VI. DESCRIPTION OF APPARATUS AND INSTRUMENTATION

Except for the following changes, the test apparatus employed for the experiments reported herein was identical to that described in references 1 and 2.

A Waterman flow control valve was installed at the coolant outlet section of the combustion chamber, and was adjusted to provide a flow

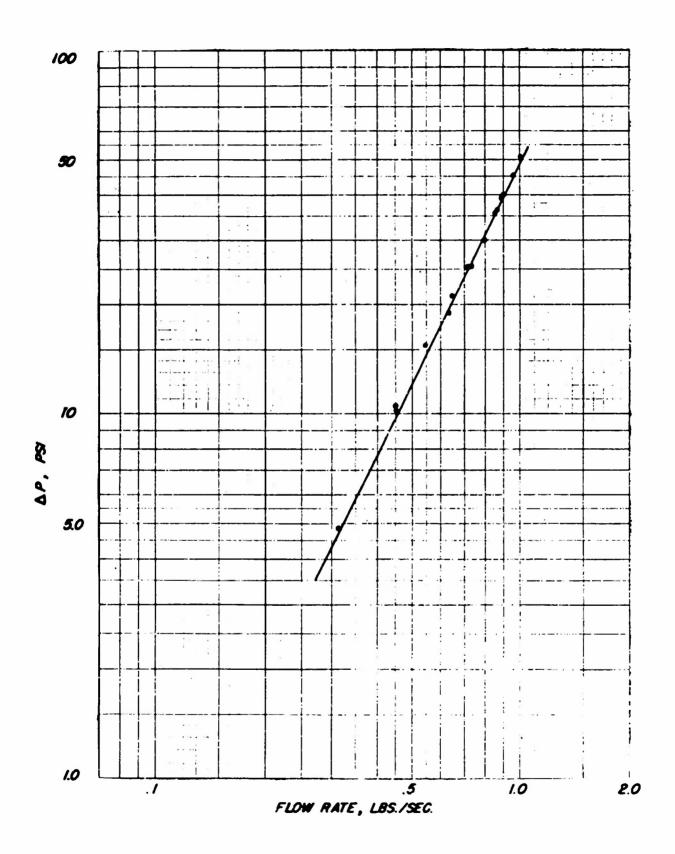
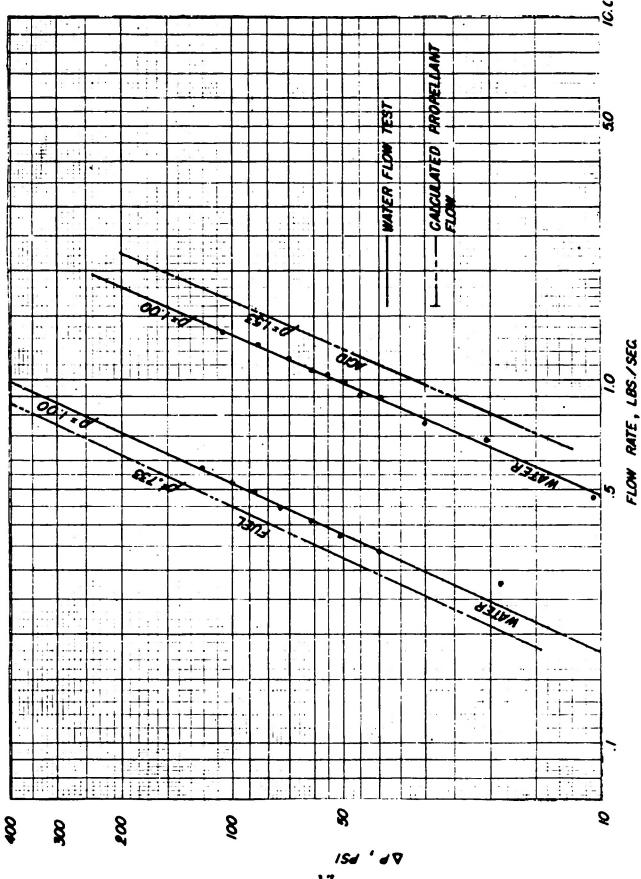


Fig. 9. Pressure Drop vs. Flow Rate for 1000 psi Turbulence Ring.



Pressure Drop vs Flow Rate for Injector 1000-2:1-12. Fig. 10.

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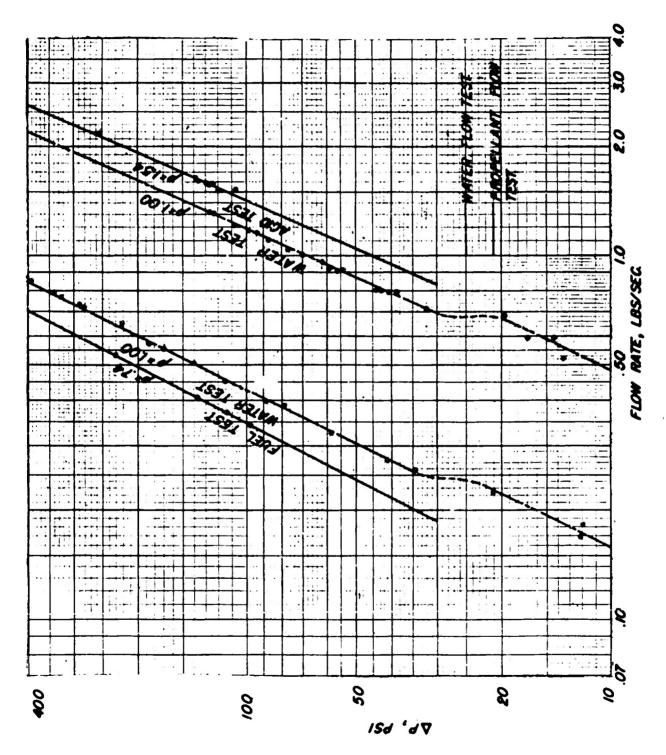
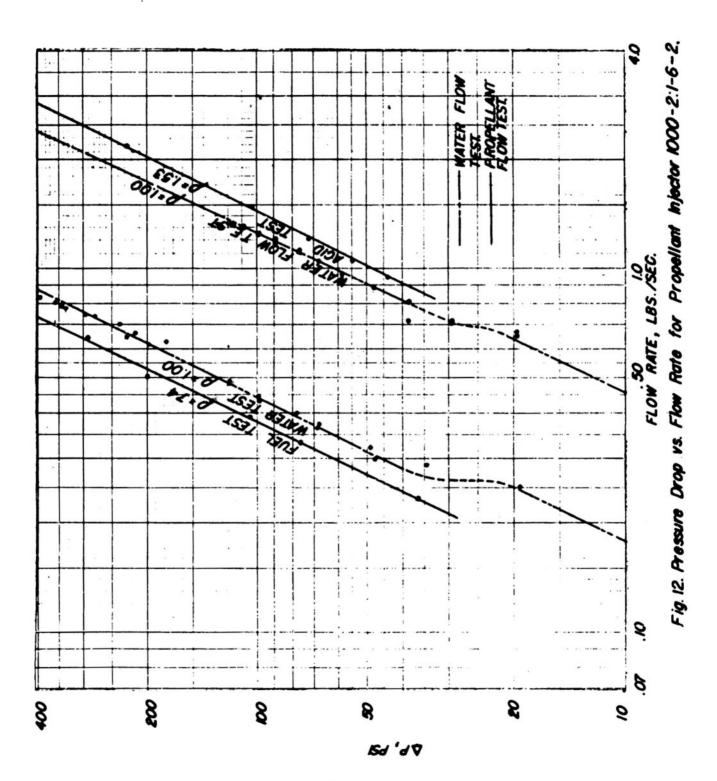


Fig. 11. Pressure Drop vs Flow Rate for Propellant Injector 1000-2:1-6-1.



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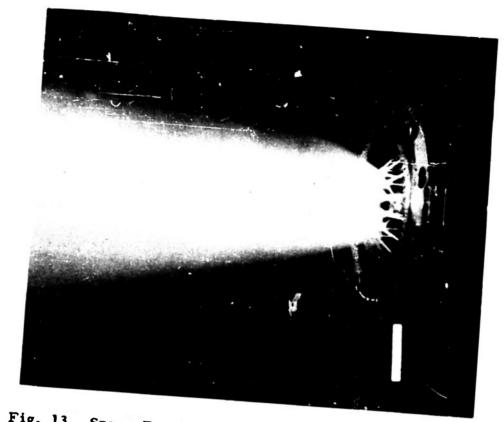


Fig. 13. Spray Pattern Produced by 1000-2:1-12 Injector

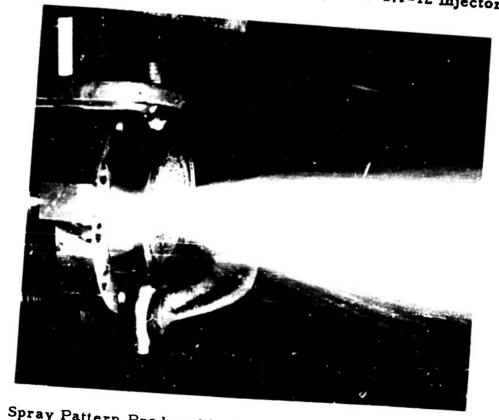


Fig. 14. Spray Pattern Produced by 1000-2:1-12 Injector and Turbulence Ring.

rate of 1 lb per sec, an inlet pressure of approximately 400 psia, and an outlet pressure of 230 psig. The coolant (water) for the combustion chamber was supplied by a centrifugal pump.

An air-pressurised cooling system was utilized for supplying cooling water to the nossle. The flow rate was adjusted to approximately 1 1b per sec by means of a hand valve connected to the nossle coolant outlet. The coolant pressure was adjusted to appreximately 1000 psig at the inlet and 200 psig at the outlet. Since the coolant supply tank was pressurised from the same compressed air supply used for pressurising the oxidiser supply tank, the nossle coolant pressure and flow rate varied slightly from one firing test to the next depending upon the propellent mixture ratio.

The hipropellant valve employed was salvaged from YEG13-AJ-5
JATO unit and was modified to provide a more positive shut off and a
slower opening speed. The pintles were redesigned in the manner previously described in reference 3.

Reluctance type transducers, described in references 1 and 3 were employed for measuring pressures, differential pressures, and thrust. The flows of coolant water to the nozzle and combustion chamber were measured by means of Flewrators, and the flew of coolant water to the turbulence ring calculated by means of Fig. 9 from measurements of the upstream coolant pressure during the run. All other flews were measured with sharp-edge orifices in conjunction with reluctance type differential pressure transducers. A more detailed discussion of the instrumentation and testing procedures employed is presented in reference 3.

#### VII. DISCUSSION OF EXPERIMENTS

Runs 172 to 180 were made primarily to obtain information concerning the operating characteristics of the entire system. The first run (Run 172) at 1000-yein combustion pressure was made with an uncooled moter (steel combustion chamber and copper nossle) to obtain starting-up characteristics. The motor started smoothly, but a miner explosion occured

on shut-dewn. For run 173, the water cooled combustion chamber was employed. The combustion chamber performed satisfactorily, but the unceeled copper nossle burned out in less than 5 seconds. A water-cooled nossle was installed and the moter was run at 700-peia combustion pressure. The nossle burned out at the threat after 35 seconds of operation. Judging from the position of the scale deposit in the coolant passage, the "burn-out" was caused by boiling of the coolant. Runs 175 to 179 were made with the ceramic motors to obtain more complete information on operating characteristics. All of the ceramic nossles burned out at the threat in less than 10 seconds. In most instances the burnout started in less than enessed. Figure 15 is a phetograph of one of those nossles after firing showing the burnout as viewed from the divergent end of the nossle.

A water-cooled cepper nossle (WG-Ou-1) was installed for Run 180. After 3 sec of running the nossle coolant flow rate decreased and the motor was shut down. Choking of the coolant flow due to the generation of steam in the coolant passage appeared to be the cause of the decrease in the coolant flow rate.

A larger coolant flow rate was employed for Run 181. The meter performed satisfactorily for 24 sec. After the shut down, however, a small hole was discovered in the nozzle just downstream from the throat. Disassembly of the nozzle showed that the high coolant pressure had forced the copper nozzle away from the filler block, and had allowed the coolant to by-pass the coolant passage. Evidence of boiling was present in the region of the burnout. The coolant passage of the latter nozzle was formed by eilver soldering a copper wire around the copper nozzle.

For Runs 182 to 187 a copper nossle (WC-Ou-2) having machined coolent passages and a uniform wall thickness (0.15 in) was employed. That nossle performed satisfactorily for run durations up to 39 sec. After Run 187 the nossle had become deformed due to the high coelent pressure to which it had been exposed. It was replaced by a new nossle (WC-Ou-3) which was designed with a tapered wall to improve its ability to resist collapse when exposed to high coolant pressures. The deformation occurred

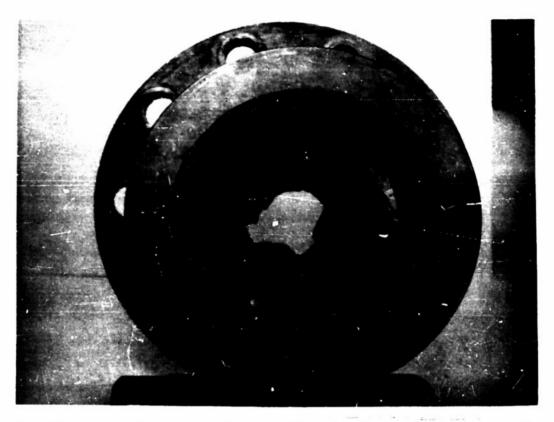


Fig. 15. Divergent Section of Ceramic Nozzle Showing Burnout at Throat

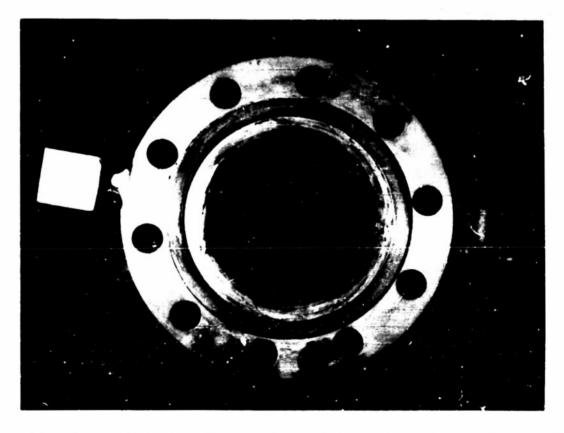


Fig. 16. 6-point Injector Showing Burned Area Around Injection Orifices

principally at shut-down when the nossle was hot, and was unsupported by the gas pressure.

The thickness of the wall was increased to 7/16 in at the combustion chamber end, tapered uniformly to 0.212 in at the threat, and then increased uniformly to 0.25 in at nossle exit. A 1/16 in deep costant passage was machined in the outer wall of the nossle. The nossle assembly included a split filler block, equipped with springs, for holding the block firmly against the lands of the coolant passage. Figure 6 illustrates the nossle assembly. Satisfactory operation of the nossle design presented in Fig. 6 was obtained throughout the remainder of the investigation (Runs 188 to 204). The above mentioned design did not totally prevent the copper nossle from creeping away from the filler block. After approximately three or four runs it was generally necessary to force the copper lands back against the filler block. This was done by driving two iron cones simultaneously into the converging and diverging sections of the nossle.

A 12-impingement point triplet-type injector was employed for Runs 172 to 187. Slight burning of the injector face in the region of the fuel holes occured during each run. Moreover, the small fuel holes employed with the aforementioned injector tended to plug up with carbon; the small holes were required to produce a reasonable pressure drop acress the fuel holes. Whether the plugging occurred during the starting-up period, during the run, or at shut-down was not determined.

To overceme the plugging of the fuel holes encountered with the 12-point injector, the latter was replaced by a 6-point triplet injector (2:1-57-1). The latter was designed to provide a higher pressure drop across the injection orifices and better cooling of the downstream face of the injector. Injector 2:1-67-1 was employed for Runs 189 to 195. Figure 16 shows the injector face after it had been used for 7 runs. A small amount of burning occurred principally around the inboard acid erifices. The burning appeared to stop after it had progressed to a depth of approximately 0.05 in, leaving a wall thickness of approximately 0.1 inch.

An explosion caused by leakage of the starting fuel (furfurylalcohol mixed with aniline) into the acid manifold occurred on Run 195,
and the injector manifold section was damaged beyond repair. The manifold section was replaced with another of identical design. Measurements
of the pressure drops across the acid and fuel holes of the new injector
showed them to be semewhat smaller than those for the previous injector
for the same flow rates (see Figs. 11 and 12). A bipropellant valve salvaged from a THL13-A J-5 JATO unit was modified to prevent leakage of the
propellants into the injector manifolds by including 0-ring seals in the
pintles. In addition, the Teflon seats were replaced with aluminum seats.
The redesigned bipropellant valve operated satisfactorily for the remainder
of the investigation, Runs 196 to 204.

JP-3 was used as the fuel for Runs 172 to 193. For Runs 194 to 204 inclusive the fuel was JP-4. We distinguishable differences were observed in the starting-up, the combustion, and the perfermance characteristics of JP-3 and JP-4.

#### VIII. EXPERIMENTAL ERROR

The calculations of performance and heat transfer for the rocket motors were made on an IBM card program computing machine in the manner previously described in reference 3.

The method employed for estimating the accuracy of the experimental performance is also discussed in detail in reference 3. Employing the customary differential calculus method for small errors, the expected maximum error in the performance parameters were calculated for
Run 204 by means of the IEM computer, the errors being 1.5 per cent for
the specific impulse, 1.4 per cent for the thrust coefficient, and 1.9
per cent for the characteristic velocity. Though the errors vary slightly
from one run to the next, they are of the same order at those given above.

For the data employed in Figures 1 and 2, the maximum deviation of the combustion pressure from 1000 psia was 2.9 per cent; however, the average deviation was 1.0 per cent. The maximum scatter of specific

impulse due to the deviation of combustion pressure is from equation 2 (see also Fig. 4)  $\stackrel{+}{=}$  0.9 sec and the average scatter is 0.3 sec for a mixture ratio of 4.6. Thus the effect of the deviation in combustion pressure from 1000 psia is within the accuracy of the experimental measurements.

Figures 1 and 2 de not include the runs which are marked with an asterisk in Table 1, Appendix A. Some of the runs so marked were utilized only for testing the starting-up characteristics of the motor. In others, instrumentation or mechanical failure in the system produced unreliable information. It may be noted in the comments of Table 1 that instability was encountered during the runs which are marked with a double asterisk. In many instances during an unstable period of the run the variation of propellant flow rate and coolant temperature was so large and so rapid that the respective recording instruments could not respond accurately. Figure 17 shows the oscillegraph record of Run 202, and illustrates the transition from unstable to normal combustion. During the run the combustion alternated to and from unstable operation several times with ne definite period of transition. For comparison a smooth run is shown in Fig. 15.

The inability of the recording potentiometer to record the pressure drep across the metering orifice employed for determining the propellant flow rate when unstable combustion was encountered, is illustrated in Fig. 19. Figure 19 shows the trace of the exidisor metering orifice pressure drop obtained from a stable run (Run 204), and the trace from an unstable run (Run 202). The stable and unstable periods can be detected easily by comparing the trace with the oscillograph record in Fig. 17. During stable operation (normal combustion oscillation) the trace is relatively flat, but during unstable operation the trace wanders or drifts. Judging from the oscillograph record, particularly that of the nozzle coolant temperature (potentiometer trace not shown), the trace preduced by the recording potentiometer because of its drifting character does not indicate the correct average value of the flow rate, but rather indicates the inability of the recorder to assume an average value when there are combustion vibrations.

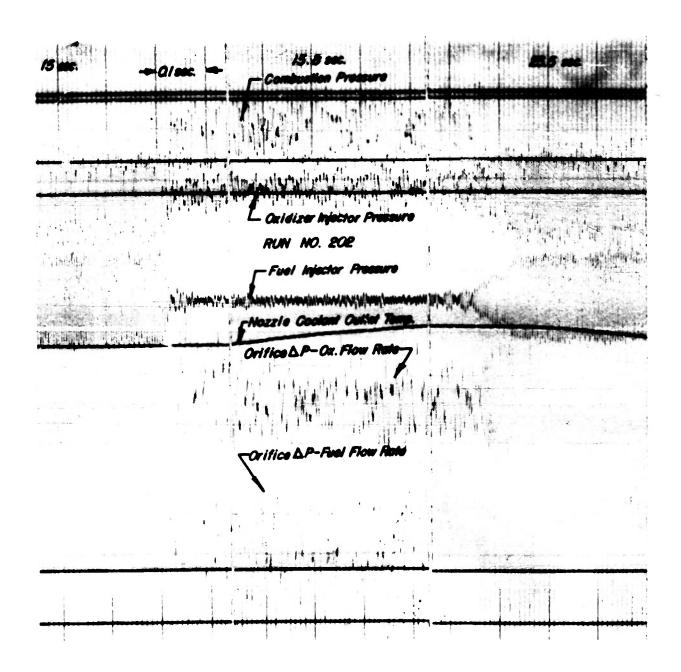


Fig. 17. Oscillograph Record of Unstable Run.

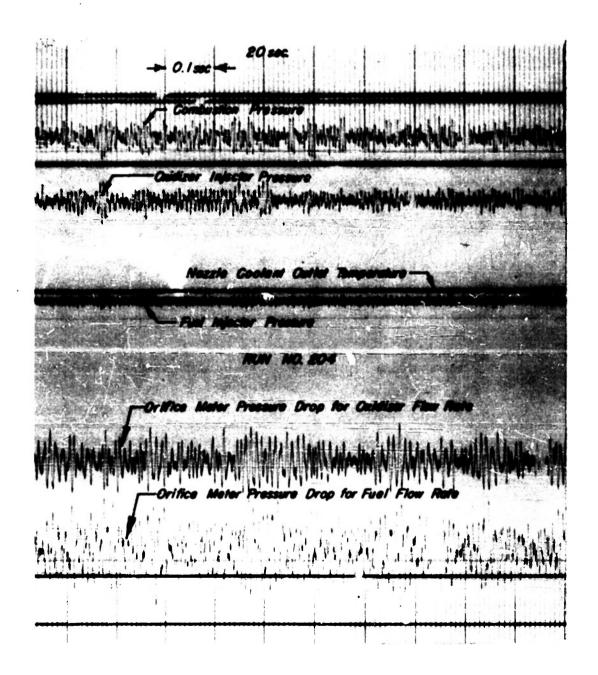


Fig. 18. Oscillograph Record of Stable Run

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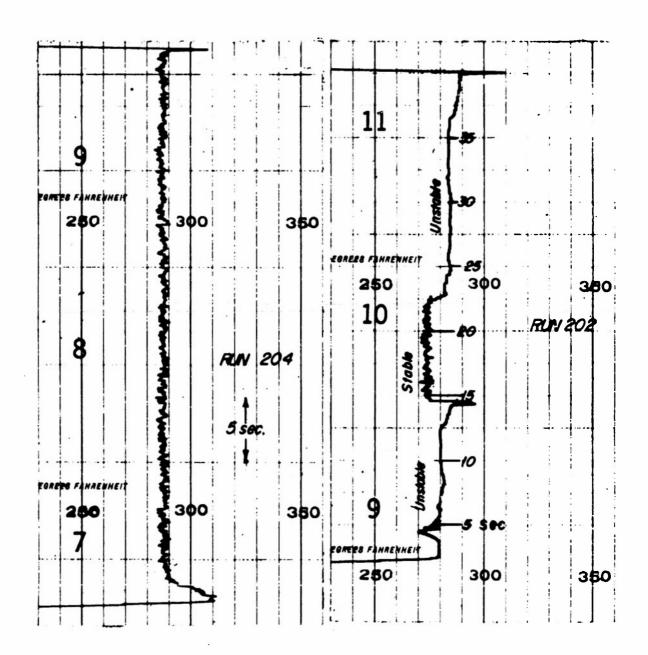


Fig. 19. Recording Potentiometer Trace of Pressure Drop for Acid Flow Rate Orifice Meter Illustrating a Stable and Unstable Run.

#### LIST OF SYMBOLS

A = surface area of combustion chamber

 $A_n =$ surface area of nozzle

C = constant

C, = thrust coefficient

0° = characteristic velocity

F = thrust

G = exidiser flew rate

G = fuel flow rate

G = total propellant flew rate

I = specific impulse

I = specific impulse corrected for heat transfer

m = constant

O/F= ratio of exidiser flow rate to fuel flow rate

p = combustion pressure

 $q_{\alpha}^{-}$  = combustion chamber heat transfer rate

 $q_n = nossle heat transfer rate$ 

q<sub>tr</sub>= turbulence ring heat transfer rate

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